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# A Review of Australian and New Zealand Investigations on Aeronautical Fatigue During the Period April 1995 to March 1997

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*Colin Martin*

**Airframes and Engines Division**
**Aeronautical and Maritime Research Laboratory**

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**ABSTRACT**

This document is for presentation to the 25th Conference of the International Committee on Aeronautical Fatigue scheduled to be held in Edinburgh, Scotland on 16th and 17th June 1997. A review is given of the aircraft fatigue research and associated activities which form part of the programs of the Aeronautical and Maritime Research Laboratory, the Civil Aviation Safety Authority, and universities in Australia and the Defence Scientific Establishment and the University of Auckland in New Zealand. The review summarises fatigue-related research programs as well as fatigue investigations on specific military and civil aircraft.

**RELEASE LIMITATION**

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## Executive Summary

The Australian Delegate to the International Committee on Aeronautical Fatigue (ICAF) is responsible for preparing a review of aeronautical fatigue work in Australasia for presentation at the biennial ICAF conference. The report, which is of interest to a number of organisations in Australia and New Zealand, is published separately as a technical note and later forms a chapter of the ICAF conference minutes published by the host nation. The format of this technical note reflects the ICAF requirements. AMRL has traditionally provided the Australasian delegate to ICAF and has been responsible for compiling and publishing the biennial review.

## Contents

## 8.1 INTRODUCTION

This review of Australia and New Zealand work in fields relating to aeronautical fatigue in the 1995 to 1997 biennium comprises inputs from the organisations listed below. The author acknowledges these contributions with appreciation. Enquiries should be addressed to the person identified against the item of interest.

AMRL Aeronautical & Maritime Research Laboratory, GPO Box 4331, Melbourne, Victoria 3001.
CASA Civil Aviation Safety Authority, PO Box 367, Canberra ACT 2601.
DSE Defence Scientific Establishment, Auckland Naval Base, Auckland, New Zealand
RMIT Royal Melbourne Institute of Technology, GPO Box 2476V, Melbourne, Victoria 3001.
University of Auckland, New Zealand

## 8.2 FATIGUE PROGRAMS ON MILITARY AIRCRAFT

### 8.2.1 F/A-18 International Follow On Structural Test Project ( D Graham AMRL)

In support of the F/A-18 International Follow on Structural Test Project with Canada (IFOSTP), AMRL is conducting two fatigue tests, a stand alone test of an F/A-18 centre fuselage wing carry-through bulkhead in support of the Canadian centre fuselage test FT55, and the F/A-18 full scale aft fuselage and empennage fatigue test, FT46. As well as the structural testing programs, AMRL staff have also been involved with a flight test program in support of the F/A-18 wing fatigue test FT245, which is being developed at IAR, NRC, Ottawa Canada. The RAAF Aircraft Research and Development Unit (ARDU), Edinburgh, South Australia has been conducting a wing loads flight test program which is providing calibrated wing manoeuvre loads data and wing buffet loads data to Canada to assist in the development of the wing test loads. This report covers the FT46 test, and the flight test work at ARDU. A separate report is included for the FS488 bare bulkhead test.

***FT46 aft fuselage and empennage fatigue test***

The testing is being conducted in two phases, which cover the periods before and after an aerodynamic fence was fitted to the wing leading edge extension (LEX) to reduce the magnitude of vortex induced buffet at the fins. In phase 1, 1250 flights (5 blocks) of flight loads from early usage containing severe fin dynamic loading will be applied, followed by the loading from the post-LEX fence usage with more benign buffet loading. At the spectrum change over point the test article will undergo a structural modification program in which fleet modifications common to the RAAF and CF fleets will be installed. These will consist of the installation of doublers on the fins, replacement of fin fasteners, and the change of engine mount support structure.

763 Spectrum Flight Hours (or 750 flights) of pre-LEX fence loading has been applied to the test article. It is anticipated that the pre-LEX fence loading phase will be completed by the end of May 1997. The test began cycling in Feb 1996. The first block of loading (250 flights) was not completed until 28 August 1996. However, the last block, the third, was completed in 14 days.

When cycling, the test can achieve an average manoeuvre cycling rate of over 0.7 load cycles per second, and although the current load sequence is over 130,000 lines long, the 250 flight block can be completed within about 40 hours of test cycling. While the percentage of time spent cycling during the first three blocks has been quite low, the rate of testing is expected to increase substantially during the post-LEX fence phase. The first block of loading was very slow because of the large number of times test limits (load, acceleration, displacement) on the advanced FT-46 combined manoeuvre and buffet fatigue test rig were exceeded. The test limits were initially set very conservatively, and were cautiously relaxed throughout the first block of testing. Testing was also halted after the first 10 and 70 flights and after every 250 flight block to conduct a detailed dynamic loads review to compare achieved dynamic loading response against target values. Plots of predicted versus achieved RMS values of acceleration and strain, power spectral density plots and exceedance curves of acceleration and strain were produced to ensure that acceptable dynamic loading was being applied before testing was continued.

The static and dynamic strain surveys carried out prior to test cycling showed that the strain distributions in the fin stub frames closely matched the strain distributions measured during flight testing, indicating that the new combined methods of test loading for manoeuvre and buffet loading closely simulated flight conditions.

The pre-LEX fence fin buffet loading is severe and minor defects, as experienced in the F/A-18 fleet are expected during the pre-LEX fence testing phase. Inspections of FT46 to date have revealed several fastener head failures in the root region of the fin representative of fleet failures.

During routine cycling 112 channels of 606.06Hz sample rate data (to match flight test data sample rates) and 490 channels of data sampled at each manoeuvre load line are collected and archived on

CDROM. Each block of testing requires over 60 CDROM's. To aid test data analysis, a reduced data set is also being generated containing turning point data for a reduced number of measurands.

The two phases of testing has necessitated the development of two separate test load sequences, utilizing flight test data from the two different fin buffet flight regimes, pre-LEX fence flying and post-LEX fence flying. Development of the post-LEX fence sequence has not yet been completed. Because engine mount support structure is being changed on the test article after the completion of the pre-LEX fence testing phase, the accurate simulation of engine mount dynamic loads will be included for post-LEX fence testing, and if necessary two additional engine loading shakers will be incorporated into the test loading system.

***F/A-18 Wing Load Flight Testing.*** (RAAF Aircraft Research and Development Unit (ARDU) Task 166)

Task 166 was established to provide wing loads data in support of the IFOSTP wing test being developed in Canada. Previous ARDU flight testing provided data in support of the centre fuselage test FT55 and the inner wing region, but there were insufficient measurands to define outer wing loading, in particular the wing fold region. As a consequence Task 166 was established to provide the additional outer wing and control surface loads not previously collected, including outer wing and control surface buffet data.

ARDU aircraft A21-032 was instrumented with strain gauges, load bridges and accelerometers and an extensive wing loads calibration program was carried out utilizing the test rig developed for the earlier IFOSTP flight test program. Wing control surfaces were instrumented and calibrated at Aerospace Technologies of Australia (ASTA) Fishermens Bend Melbourne. Calibration equations have been established and load verification flights have been completed. Flying of the 60 hour flight test matrix, developed to cover both RAAF and CF flying conditions, has begun.

### 8.2.2 F/A-18 IFOSTP FS488 Bare Bulkhead Fatigue Test, FT488/2 (Michael Houston - AMRL)

The F/A-18 FT488/2 Bare Bulkhead Fatigue Test, at AMRL, is being conducted in support of the Canadian centre fuselage portion (FT55) of the F/A-18 International Follow-On Structural Test Project (IFOSTP). The objectives of the test are to determine the safe life of the FS 488 bulkhead under RAAF/CF usage (represented by the loading spectrum being applied to FT55), and to assess rework options for the FT55 FS 488 bulkhead if full life is not achieved. Routine cycling commenced on 4 December, 1995, and test completion is anticipated for mid-June, 1997. The test has completed 20,000 Spectrum Flight Hours.

As the aft most of three wing carry-through bulkheads on the F/A-18, the FS 488 bulkhead provides a primary load path for wing bending, and to a lesser degree wing torsion and shear. Various configurations of the bulkhead have been subjected to many fatigue tests and strain surveys, both as a stand-alone component and as part of the built-up structure. These were conducted to support redesigns of the flange profile and drag longeron recess located directly below the lower wing attachment lug and culminated in an improved configuration, now found on most F/A-18 A/B and C/D aircraft around the world. Failure of this configuration is characterised by widely distributed, multi-site cracking, suggesting an optimised design. The FS 488 bulkhead is generally still considered to be one of the most fracture critical components in the airframe due to the short critical crack length (6mm), access difficulties (for inspection), and the importance of the part to flight safety. The FT488/2 test brings together this current configuration and the RAAF/CF usage spectrum, an untested combination.

The FT488/2 test rig subjects the bulkhead to a program of upper and lower wing attachment lug loads, applied symmetrically. Although the entire bulkhead is loaded, only the particular section of the Outer Mould Line (OML) below the wing lug attachment point is considered to be representative due to the absence of the supporting structure and other significant load paths found in the built-up aircraft (eg. skins, longerons). To prevent out-of-test-area failures, the lower portion of the bulkhead (i.e. main

landing gear bay and surrounding structure) is restrained by a fore-aft bracing strap assembly. Strain response in the test area on FT488/2 closely approximates that on FT55 for any given loading condition.

The test article is fitted with several types of instrumentation. The starboard OML flange was heavily strain gauged to verify loading and provide control points common with FT55. A collection of far field gauges enable monitoring of structural integrity elsewhere on the bulkhead. The AEMS-1001 acoustic emissions system, which relies on two transducers, is installed within the port test area. This equipment is also being utilised on FT55. Prior to the 58th program, a new crack detection/monitoring facility was fitted on the port flange. The "Davey System", produced by Structural Monitoring Systems of Perth, comprises a pressure transducer, digital display, vacuum pump, and a silicon pad or pads molded to cover the region selected for monitoring (the port flange outboard, forward and aft faces on FT488/2). Each pad features an array of channels which is sealed against the surface of the structure during installation. Alternate channels are open to ambient pressure, while the rest are held under vacuum. The pressure transducer measures the differential between these two sets of channels. Consequently, since the silicon wall between any two channels is only 250mm wide, a crack of this length or greater will register as a cyclic change in pressure differential. Sensitivity is primarily dependent on the geometry of the pad, and, for this application, the system offers a potential improvement of 200% or more over conventional NDI, particularly on a shot peened surface.

Following the application of 57 programs, representing 18,520 Spectrum Flight Hours (SFH), a 1mm crack was discovered in the forward 6 inch radius of the starboard OML flange. The defect was detected using a 500KHz-1MHz Eddy Current Probe, and confirmed with Fluorescent Penetrant under load. Earlier surface preparation (abrasion of the shot peened surface) to facilitate strain gauging in this region may have negated the beneficial effects of shot peening in the region. This issue has not yet been resolved; however, to allow the test to continue, the cracked material was excised for fractography, and the edge blended, polished, and glass bead peened (See 8.2.5). Such a rework, which removes fatigued material at a predetermined point in the life of the bulkhead, is under consideration for its value as a viable life extension technique.

Since the rework of the starboard flange, another five programs have been applied to the test article without any further crack indications from acoustic emission, the "Davey System", or eddy current and dye penetrant NDI.

### 8.2.3 Cracking in F/A-18 Bulkhead Fatigue Test (G Clark, S Barter and M Houston- AMRL)

Fatigue cracking was detected in a stand-alone F/A-18 bulkhead fatigue test, and the location of the cracking — in an area which had been peened for life extension, but then abraded to allow strain gauging — highlights the difficulty in interpreting fatigue test results which contain non-representative areas. The question of whether the result indicates a representative fatigue life is being addressed by machining out the crack and re-peening to achieve enough life to allow failure to occur elsewhere. Initial quantitative fractography of the cracking has revealed an unusual crack morphology, whereby the crack has grown more rapidly into the material than along the surface. The unusual crack profile was not consistent with cracking observed in similar test articles; this is due to the interaction of residual stresses from peening and gross crack branching.

### 8.2.4 F/A-18 Coupon Fatigue Testing (W Madley - AMRL)

**Spectrum Coupon Tests**

As part of the F/A-18 International Follow On Structural Test Project (IFOSTP), eight fatigue-critical regions have been identified on the aft fuselage and empennage, and coupon fatigue tests are planned to simulate several of these regions. Earlier work concentrated on the vertical tail stub frame region and the horizontal stabilator spindle under loading typical of early Australian and Canadian usage (including the high dynamic loading environment prior to the fitment of the Leading Edge eXtension (LEX) fence). It was clear from this testing that the buffet loads dominate the fatigue life for the

vertical fin, but for the stabilator spindle the buffet and manoeuvre loads are equally damaging. Current testing simulates the same regions of the vertical tail and horizontal stabilator, but under loading typical of current usage, including the reduced dynamic loading environment after the fitment of the LEX fence. Specimens simulating the vertical tail root demonstrated that while buffet loads still dominate the fins, they are significantly less than those generated prior to the fitment of the LEX fence.

Fatigue life predictions for all test sequences are being made and compared with experimental lives. The main reason for these comparisons is to gain confidence in the use of life predictions which will assume some importance once a full-scale article failure has been obtained. Both crack initiation and crack growth models are being used, details of which were given in the previous review.

### 8.2.5 Materials Testing (Constant Amplitude) (W Madley - AMRL)

The IFOSTP fatigue analysis group is also undertaking a basic materials testing program to update the current material properties knowledge for aluminium 7050 for use in fatigue analysis. The main thrust of the project is to obtain strain - life information in the low amplitude / high cycle region of the life curve, ($10^6$-$10^8$ cycles), in order to study the effect of post-LEX fence buffet.
The details of the testing program are as follows:

1. Strain - life information will be obtained for up to 10 different strain amplitude levels, ranging in life from approximately $10^3$ to $10^8$ cycles.
2. Strain -life information will also be obtained at three different strain ratios, of $R_\varepsilon = -1.0, 0.0, 0.5$.
3. Cyclic and hysteresis curves will be obtained.
4. As well as tensile information.
5. A cylindrical specimen with tangentially blended fillets between the test section and ends will be used.

Validation testing for the proposed program and coupon design is near completion. A small series of tests were conducted at $R_\varepsilon = -1$ to validate the coupon design. Seven coupons have been tested under load control, with lives in the region of $\sim 10^5$-$10^8$ cycles. Another specimen was tested under strain control to obtain cyclic information, and to establish the maximum limits required during testing. From these tests, minor changes have been made to the specimen, and manufacture of specimens to this new design is proceeding.

### 8.2.6 F/A-18 Usage Monitoring ( L Molent - AMRL)

In support of the RAAF F/A-18 aircraft structural integrity program, AMRL is developing improved fatigue usage and analysis capabilities. Since the last ICAF report, work has been carried out on the existing fatigue usage monitoring program called SAFE, and on a simpler program called MSMP both of which use data from the F/A-18 on-board Maintenance Signal Data Recording System (MSDRS). Consideration has been given to identifying the sources of errors associated with using strain measurements for fatigue usage analysis. Finally work has been conducted on an alternative usage monitoring system called AFDAS. Each of these aspects are reported in this item.

***Structural Appraisal of Fatigue Effects (SAFE) program***

Fatigue usage monitoring on the F/A-18 aircraft is carried out using strain measurements recorded on the on-board data acquisition system and analysed using software derived from the NAVAIR "F/A-18 Structural Appraisal of Fatigue Effects (SAFE)" program. The basic data from this program comes from the aircraft's on-board data acquisition system known as the F/A-18 "Maintenance Signal Data Recording System (MSDRS)". The MSDRS records time based data from the aircraft's data bus and strain sensors located at fatigue critical locations throughout the aircraft.

The fatigue life monitoring program (SAFE) implemented for the aircraft, essentially compares the usage of an individual aircraft to that of a representative fatigue test article (i.e. subject to a similar operational load spectrum). As the aircraft was designed and certified to a safe life philosophy, when the damage accumulated on a particular aircraft matches that calculated to have been imparted to the test article at the completion of testing, with appropriate safety factors applied, the aircraft is said to have consumed its safe life. Since the structure which primarily governs the safe (or economical) life of the aircraft is currently believed to be one of the centre fuselage wing attachment bulkheads, the strain sensor placed at the wing root (WR) titanium lug at fuselage station FS470, to monitor the fatigue usage of the centre fuselage and inner wing, is the prime measurement source.

Significant differences have been noted between the fatigue damage calculated by SAFE and an alternative method based on Nz exceedances, known as the Mission Severity Monitoring Program (MSMP). A working group (WG), consisting of RAAF ASI, DSTO's Airframes and Engines Division (AED) and Hawker de Havilland Victoria Ltd (HdHV) has investigated the SAFE process in detail and has identified a number of limitations or deficiencies in the following areas:

The MSDRS parameter data checking routines; the calibration of damage models to durability tests; the reliability of strain sensors; the procedures for data fill-in for missing sensor data; the in-flight calibration of strain sensors; the treatment of the effects of overloads on life predictions; the strain based sequence accountable cumulative fatigue damage model, and the available coupon test and material data. For the RAAF mean usage spectrum, the fatigue lives computed by the SAFE program were shown to be unconservative. Amendments to the current SAFE process were also addressed.

The findings of the working group are presented in Ref [1], and a summary of some of the lesson learnt from experience with the MSDRS fatigue monitoring system is presented in Ref [2].

***Mission Severity Monitoring Program***

The Mission Severity Monitoring Program (MSMP), has been developed by the RAAF, AMRL and HDEH as a simpler, quicker, and more user-friendly program than SAFE to determine the relative usage severity for a mission for use by operational squadrons. For comparison, SAFE uses crack-initiation models to predict the damage induced by each cycle in a sequence dependant process that takes account of the existing state of stress and strain at a notch. MSMP uses an average damage value for a cycle of known amplitude and mean strain to predict the life. These averaged damage values, developed for a series of means and amplitudes, constitute an unit damage matrix (UDM), which is used to arrive at a fatigue index. Figure 1 outlines the operations carried out in MSMP and Ref [3] presents a review of the current versions of the MSMP used by the RAAF. It summarises the

research carried out to determine the validity of the damage model used, recommends improvements to the current software, and checks the accuracy of its implementation against fatigue test coupon results.

### *Strain Gauge Sensor errors*

Because both the SAFE and MSMP programs use strain measurements and determine fatigue usage by relating the measurements to the damage accumulated on a fatigue test article then it is important that the strain sensors are calibrated such that loading derived from these can be related directly to the equivalent load on the fatigue test article. Two gauges placed at nominally identical locations, but on different airframes, may produce varying responses to nominally identical loading, due to: slight differences in airframe build quality, slight gauge alignment differences and variations in gauge factors or gauge sensitivity. Multiple load paths in redundant structure may also cause varying gauge response, where differences between aircraft may be "built-in" before delivery. Differences will also arise due to differences in zero strain setting, ie. 1*g* level flight or pre-take off ground condition, or due to sensor drift or sensor non-linear response. Because it is often impractical to calibrate every sensor in the fleet using a ground load calibration, it is normal to rely on in-flight recorded data to relate gauge response between aircraft to the loads on calibrated aircraft. To do this, data recorded at unique Points In The Sky (PITS) which give reproducible values of the principal loading actions are required. For some sensors, namely for gauges on the wing, this is a relatively simple procedure and is described in Ref [1].

In other instances, for example vertical and horizontal tails, the calibration process requires more parameters in order to define unique PITS. Thus a monitoring system which records parameters other than strain is essential to enable scaling factors to be determined. In the case of the F/A-18 Horizontal Tails (HT) and Vertical Tails (VT), analytical techniques have been determined Ref [4, 5]. Ideally any analytical calibration techniques such as those discussed above should be validated by conducting aircraft ground calibrations on a sample of aircraft prior to routine use. A limited fleet static calibration exercise is currently being undertaken by AMRL on a sample of RAAF F/A-18 fleet aircraft.

### *Correlation of AFDAS with MSDRS Strain Data*

All RAAF F/A-18 aircraft are fitted with the DSTO pioneered Airframe Fatigue Data Analysis System (AFDAS), manufactured by British Aerospace Australia. The AFDAS collects and processes aircraft strain and accelerometer sensor data, providing additional monitoring capability to the MSDRS sensors. Of particular importance is its capability to monitor the strain distribution across the three fatigue critical wing attachment bulkhead locations. In Ref [6] the integrity of operational F/A-18 AFDAS data was investigated, by comparison with simultaneously recorded MSDRS data. Not only was the AFDAS data validated but a potential analytical AFDAS strain sensor calibration process was established. Once data integrity can be routinely assured, the AFDAS has the potential to become a powerful and yet simple fatigue life management tool since it compresses usage data in a simple Range Mean Pair (RMP) format and also has the potential for high rate sampling capable of registering buffet loading.

### 8.2.7 PC9/A Fatigue Test (I. Anderson - AMRL)

DSTO is conducting a full scale fatigue test of a Pilatus PC9/A airframe which is representative of the RAAF fleet. The aim of the test is to define a validated service life of the PC9/A in accordance with accepted airworthiness requirements. Test loads representative of those experienced in flight are applied to the fatigue test article. Loads are applied to the wings, tailplane, elevator, fin, fuselage, engine mount frame and main landing gear, via a system of closed-loop servo hydraulic jacks and fixed reactions. A digital controller/data acquisition system is used to control the loads applied to the test article and to gather data from strain gauges installed at various locations on the airframe.

A PC9/A fleet aircraft was instrumented to measure flight parameters and flight loads. The instrumentation included an extensive suite of strain gauges at key locations on the airframe. A comprehensive flight test program was conducted, and flight strain and flight load histories representative of fleet operation were recorded. The sequence of jack loads applied to the test article is based directly on flight load histories.

Prior to commencing the fatigue test, extensive commissioning and test loads verification work was performed. During this stage (the period July 1995 - 14 January 1996) a number of trial loadings were conducted, which included complete flights, partial flights and trial loading cases. The purpose of this work was to check the function of the test rig and to verify the sequence of test loads. Loads verification work included strain based relative damage checks between test recorded strain histories and flight recorded strain histories for the majority of the flights that comprise the overall test load sequence at several locations on the airframe. During the course of commissioning and loads verification a number of refinements and improvements were made to test rig loading systems and the sequence of test loads. In early January 1996 the test load sequence and test rig configuration was finalised and formal test running commenced.

At March 1997 the test article had accumulated 15,000 simulated flight hours. A number of cracks have developed in the airframe. The most significant cracking identified is in the upper surface of the wing near the wing to fuselage attachment fittings and in the fin support structure of the fuselage. The test article is scheduled to accumulate 30,000 simulated flight hours by the end of 1997 and 50,000 simulated flight hours by the end of 1998.

References 7 to 18 cover the Publications on this task. Note, many of these publications have limited release conditions attached.

### 8.2.8 Macchi MB-326H Fatigue (G Clark - AMRL)

The Macchi Recovery Program (reported in the 1995 review) represented a major effort by RAAF and DSTO-AMRL to ensure that cracking and corrosion problems which threatened the viability of the RAAF Macchi trainer fleet did not result in a need for hasty acquisition of new trainer aircraft. Under the Program —now complete— the number of aircraft in the RAAF fleet was reduced, and then maintained by the introduction of new wing sets. The basis for allowing the continued service of the aircraft through to the year 2000 was a series of investigations completed by RAAF and DSTO in which twelve wings, two fuselages, and two empennages were torn down to acquire cracking and corrosion data sufficient to support a re-lifing of the aircraft. This program, based in part on fatigue testing of centre section booms, static testing of tailplanes, and numerous crack growth analyses also led to the development of innovative methods of assessing crack growth in torn-down components; one such approach allowed assessment of the severity of stressing in various locations, permitting the identification of critical areas in several components. The aircraft is now maintained on the basis of a revised safe life, with some components subject to a safety-by-inspection approach. This topic will be the subject of a paper in the 1997 Symposium.

### 8.2.9 F-111 Lower Wing Skin Bonded Composite Repair Substantiation (K.F. Walker - AMRL)

A bonded composite repair to a critical fatigue crack in the lower wing skin of an RAAF F-111 aircraft has been fully substantiated. The crack was located in the lower wing skin at approximately 2/3 semi span (Figure 2). The crack had reduced the residual strength of the wing to below design limit load. The repair consisted of a 14 layer boron epoxy laminate patch which was bonded to the outer surface using FM-73 structural film adhesive. This is thought to be the most critical bonded repair yet undertaken anywhere in the world. The repair was designed and applied according to a RAAF document known as RAAF Standard Engineering C5033. A comprehensive substantiation program including testing of representative specimens and Finite Element Analysis was undertaken and has been successfully completed.

The Finite Element Analysis (Reference 19) included a calibrated three dimensional model which was subjected to structural and thermal loads. The analysis verified the adequacy of the original design which was performed using simplified equations based on 2-D lap joint theory. A comprehensive testing program (Reference 20) using representative cracked and patched specimens demonstrated the repair to be airworthy. The results of static load tests on representative specimens are shown in Figure 3. The tests show that the original crack had reduced the residual strength to below design limit, and that the repair restores static strength to design ultimate load capability, even at extremes of temperature. Spectrum loading tests were also performed on cracked and patched specimens. The crack continues to grow under the patch but the crack growth rate is reduced to an acceptable level as shown by the results for a range of patched and unpatched panels in Figure 4.

The results of this program have enabled the repaired aircraft to be returned to full operational status, and have given confidence in the design and application procedures for bonded repairs.

### 8.2.10 F-111 DADTA SUPPORT (K. Watters AMRL)

AMRL has recently completed work on the fuel flow vent hole number 13 (FFVH#13) stress analysis, [21]. The results of that analysis have been dispatched to the OEM (Lockheed Martin Tactical Aircraft Systems, LMTAS) as input to a durability and damage tolerance analysis (DADTA) of that location. An FE model of FFVH#13, allowing for a number of configuration shapes representative of the range of fleet rework shapes, was developed. The Stouffer-Bodner constitutive model was implemented into the PAFEC FE code [22] in order to obtain an accurate solution for residual stresses left by plastic deformation under cold proof load testing (CPLT). The Stouffer-Bodner model and its PAFEC implementation have been validated [23,24] against well defined representative specimen tests.

A full-scale static test program to simulate the CPLT loading of an F-111C wing is now complete. The strain data [25,26,27] were used to calibrate the AMRL FFVH#13 FE model. Those data were also dispatched to LMTAS to calibrate an FE model of the F-111 wing pivot fitting developed by LMTAS.

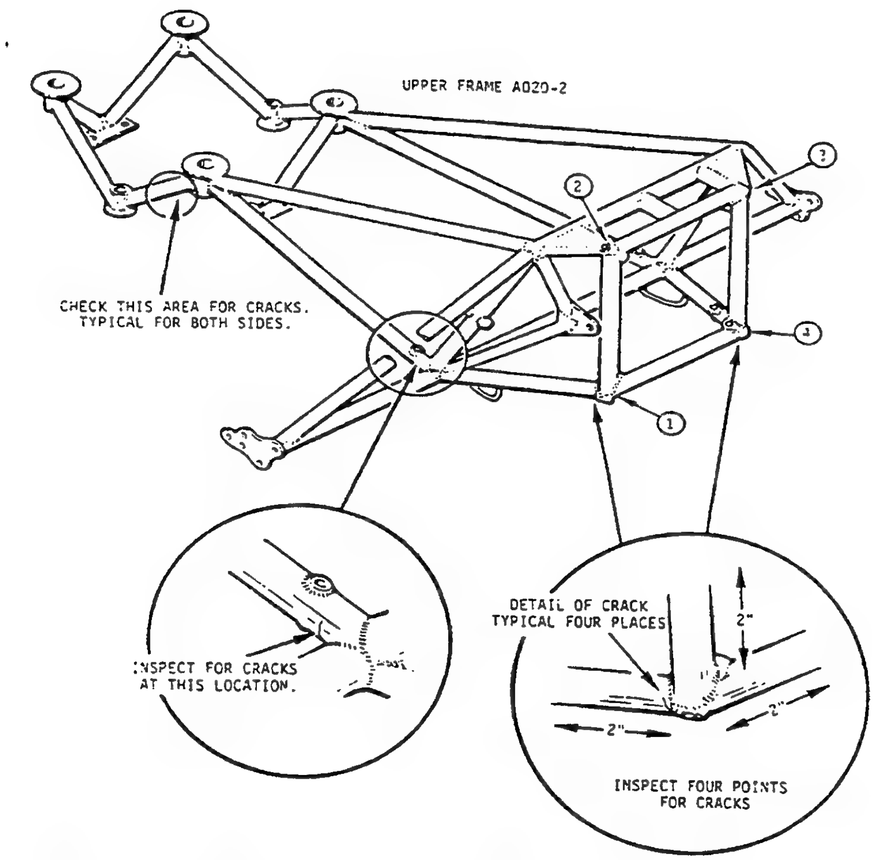
Planning is under way for a large program of F-111 structural integrity work. The work program has been precipitated by: the retirement of the USAF F-111 fleet leaving Australia as the sole F-111 operator; the anticipated diminishing OEM support; and the planned withdrawal date of 2020 for the RAAF F-111 fleet producing an ageing aircraft state. The F-111 work program will involve a comprehensive airframe audit for fatigue and corrosion, teardown inspections, loads characterisation, DADTAs, risk analyses and repair development.

## 8.3 FATIGUE OF CIVIL AIRCRAFT

### 8.3.1 Robinson Helicopter R22 Upper Gearbox Mount Frame (Mathew Richmond - CASA)

The Robinson R22 light helicopter has a tubular steel frame which supports the gearbox. This frame has a long history of cracking which highlights two common problems. Early frames cracked because of **residual stresses** at the welds. The manufacturer improved the stress relieving and prudently did a fatigue test for confirmation. But when the improved frame entered service in Australia, it still cracked well short of the published service life.

The reason this time is the use of the **wrong loads** for fatigue analysis. The cracks occur in helicopters used for aerial cattle mustering, a severe operation not accounted for in the testing or subsequent analysis. The manufacturer is designing a stronger frame for aerial cattle mustering. In the interim, Australian helicopters used for that purpose must be frequently inspected for cracks as shown in the diagram below:

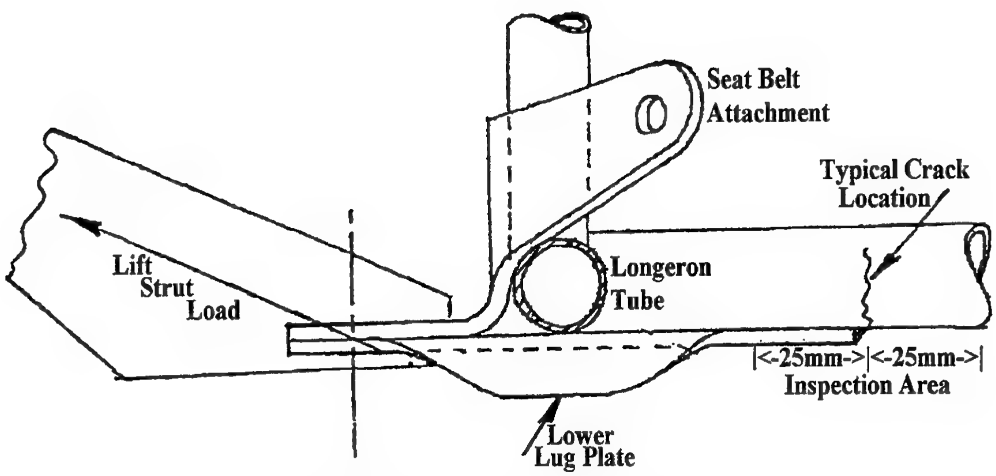



**INSPECT ALL TUBES FOR CRACKS WITH PARTICULAR ATTENTION TO THE AREAS INDICATED ABOVE**

### 8.3.2 Skyfox CA22 Wing Strut Carry - Through (Mathew Richmond - CASA)

The Skyfox CA22 is an Australian development of the popular Kitfox amateur-built aircraft from the USA. One Skyfox was nearly lost when the carry-through for the wing struts, a steel tube, broke whilst the pilot was turning onto final approach. Fortunately the aircraft remained sufficiently intact to land safely.

The failure was caused by fatigue. Although the aircraft's design standard, JAR-VLA, suggested that the stresses were too low to cause fatigue, service experience has now proved otherwise. There were three main contributing factors, none of them unique: a discontinuity; a weld; and a severe load spectrum (flight training at low altitude). This example sounds a note of caution for the use of the threshold stress levels in JAR-VLA. Australian Skyfox aircraft must now be regularly inspected as shown in the diagram below:




RIGHT HAND FITTING, VIEW LOOKING AFT

### 8.3.3 Cessna 206 Trim Tab Actuator Mounting Bracket (Mathew Richmond - CASA)

Whilst descending after dropping parachutists, the pilot felt the elevator flutter. He reduced speed, but by the time the flutter stopped it had extensively damaged the elevator trim tab and horizontal tail.

The bracket which supports the trim tab actuator had broken by fatigue. It seems that it had been cracked for some time before the final failure, its flexibility while cracked having caused the horn on the trim tab to crack. But while the cracks in the horn were detected and repaired (not long before the flutter incident), the cracks in the bracket were less obvious and were missed. With the crack in the bracket growing undetected, it was only a matter of time before it broke and the elevator fluttered.

It seems that this sort of incident has occurred before. An old Cessna service bulletin contains instructions to routinely check the trim tab actuator bracket for cracks exactly like this one. Whatever the reason, a crack which had been growing for some time was missed. Hence the current emphasis on fixing rather than inspecting known fatigue-prone areas.

### 8.3.4 Pitts Special S-2A Control Stick (Mathew Richmond - CASA)

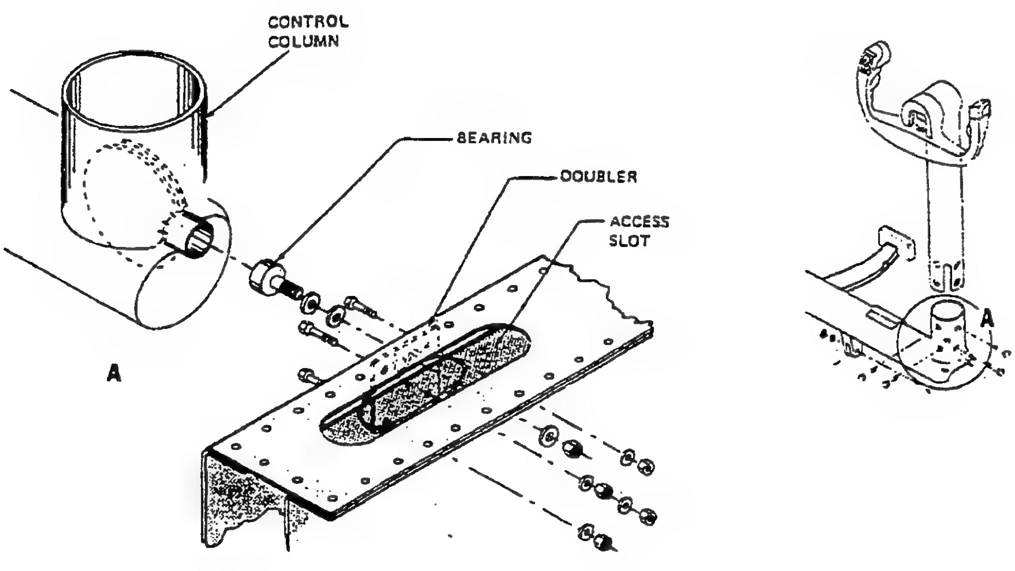
While teaching aerobatics, the instructor's control stick broke off at the base in the middle of a manoeuvre. Fortunately, the student was able to take control and land the aircraft safely. It was lucky that the front seat was not occupied by a non-flying passenger - or the forward control stick would not have been fitted!

The control stick, a steel tube, suffered from fatigue at a welded gusset - welds again! The control sticks in Australian aircraft must now be regularly inspected unless the aircraft is fitted with a new, stronger part.

This instance of fatigue highlights the need for vigilance, even on aircraft types with a long service history. Fatigue is full of surprises. This aircraft was not leading the fleet and would not generally be considered a 'high time' aircraft.

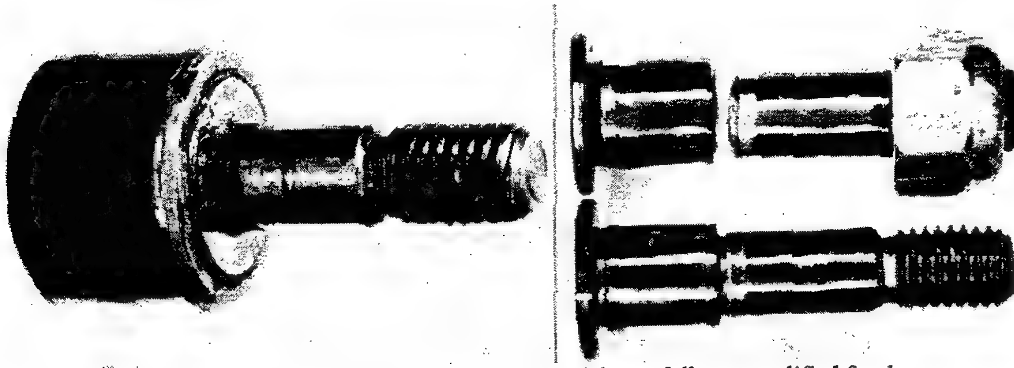
### 8.3.5 SA 226 Metro Control Column Pivot (Rick Fischer - CASA)

Late in 1996, the control column pivot of a SA 226 failed during approach to Sydney Airport. Despite having little or no pitch authority, the pilots landed the aircraft safely using engine power and flaps. Fortunately the weather was calm and clear and the aircraft's centre of gravity position was benign.




Control Column
Roller Bearings

The control column failed at the pivot bearing. The shank of the pivot bearing failed as a classic case of reverse-bending fatigue after 15,000 hours of flight.



An intact pivot bearing, which appears to be a commercial cam follower qualified for the purpose.

Top: the broken pivot bearing.

The bearings in Australian SA 226 aircraft must now be replaced before 5,000 hours of flight.

The aircraft manufacturer and the FAA have so far taken a different approach, requiring a one-time check of nut torque, having attributed the failure primarily to an under-torqued nut. The matter is currently the subject of debate. Contention surrounding the cause and corrective action for a fatigue failure will come as no surprise to experienced fatigue practitioners. With fatigue, things are never simple. Open and honest technical debate is an essential part of the fatigue business.

This and the two previous items highlight that control systems, not just airframes and engines, can be a concern with aging aircraft. Sometimes fatigue in the small parts can be just as hazardous to flight safety.

### 8.3.6 Autogyro Rotor Blade (Stephen Emery - CASA)

The rotor blade of an autogyro broke in flight, causing a fatal accident. Analysis of the failure showed fatigue. A crack had originated from an area of fretting near a bolt hole on the lower surface of the blade. The bolt hole was for the outboard bolt in the blade attachment splice plates. Blade flexure and high contact stresses inside the splice fittings made fretting inevitable.

There are three points of interest in this failure:

- the design of the joint was statically acceptable but poor in fatigue, typical for this class of amateur-designed aircraft;
- the cracking was in the blade's lower surface which, with blade droop at rest, closed the crack, preventing easy detection; and
- the splice plate itself also impeded discovery of the crack, by covering the crack until it was over an inch long.

## 8.4 FATIGUE RELATED RESEARCH PROGRAMS

### 8.4.1 Modelling of Short Crack Growth under Asymmetric Loading (C. H. Wang - AMRL)

In [28] a model of short fatigue crack growth is proposed which is based on the blocked slip concept and the shear decohesion mechanism. The analysis is extended to the case of mean stress loading. A theoretical proof is presented for the transfer of slip bands across grain boundaries. The rate of growth is proportional to the shear strain range and the maximum plastic zone size. There are no adjustable parameters in the theory for the case of high strain level, when the plastic strain dominates the decohesion process. Otherwise only one constant is needed, which may be derived from long crack growth data. The model is shown to provide satisfactory predictions of experimental results under uniaxial loading with various stress amplitudes and mean stresses.

### 8.4.2 Stress Concentrations in Bonded Lap Joints (C. H. Wang and M. Heller–AMRL)

The problem of the stress concentration in adherends of adhesively bonded joints has been analysed using a successive boundary stress correction method [29]. Joints with a square-edged adhesive layer or with a spew fillet near the ends of the overlap have been considered. It is shown that the main cause of the stress concentration stems from the adhesive shear stress which acts on the adherends; the adhesive peel stress has little effect. In the case of square-edged adhesive layer, an eigenfunction solution has been derived for the adhesive shear stress distribution near the end of overlap, which satisfies the free surface condition. Based on the improved adhesive shear stress solution, an approximate formulas has been obtained for the stress concentration factor, which is shown in Figure 5 to be in close agreement with finite element analysis. Finally strategies to reduce stress concentrations have been discussed.

### 8.4.3 Fatigue Life Prediction of Cracked Plates with and without Bonded Patch (C. H. Wang-AMRL)

This investigation aims to assess the capabilities of a NASA developed fatigue crack modelling software, FASTRAN, in predicting the crack growth rate results obtained from panels made of aluminium alloy with a thickness of 3 *mm*. The original tests were conducted by the National Aerospace Establishment, Canada, using specimens manufactured by AMRL under FALSTAFF spectrum. As shown in Fig.6, the prediction of FASTRAN with the constraint factor taken to be 1.732 exceeded the experimental results by a factor of three.

It has been found that FASTRAN does not have the capability of dealing with patched cracks. Work has since been undertaken at AMRL to develop a crack closure model for patched cracks. As shown in Fig.7, the prediction based on the new model is in good agreement with the experimental results.

### 8.4.4 Effects of Corrosion on Structural Integrity (G Clark and G Cole - AMRL)

A report has been completed [30] reviewing the effects of corrosion on structural integrity. The report was initiated in response to perceived difficulties in assessing structural integrity in cases where RAAF aircraft has suffered corrosion damage [ref.: section 9.2.3, in 1995 Australian ICAF review], and assessed world literature, documenting progress and activity in a range of corrosion/structural integrity areas. It was concluded that there is no evidence of a single underlying principle which will unify the various observed effects, and therefore research will need to progress on a number of fronts, there are several avenues of research (long and short-term) which, in addition to developing an improved understanding of the structural implications of corrosion defects, have the potential to assist in the decision making process required when corrosion defects are discovered in critical airframe components. A crucial question is the development of a means of describing the effectiveness of corrosion defects as initiators of fatigue cracking, and the approach selected involves the development and testing of a means of describing passivated corrosion in terms of an Equivalent Initial Flaw Size

(EIFS). Other areas selected for attention include the development of models which will allow analysis of the behaviour of corroded joints, and the development of an understanding of the capability of laminar corrosion defects to initiate fatigue cracking. Further work is planned to address these and other aspects of the problem.

### 8.4.5 Quantification of Peening Effects in Fatigue (G Clark and P K Sharp - AMRL)

The use of peening to extend the fatigue life of aircraft components is well established, although work at AMRL has highlighted the problems (including life degradation) which can be associated with inappropriate peening of high-strength aircraft aluminium alloys. Peening is used extensively in aircraft such as F/A-18, and in view of the need for life-extending repairs in fatigue tests, estimation of the life extension factor associated with peening is essential. Recent and current work is investigating the effects of a range of peening parameters on the fatigue performance of an F/A-18 aluminium alloy. The tests include an evaluation of the peening run-out region, and the effect of peening at corners. One aim of this work is to optimise the peening parameters for this alloy to ensure maximum life extension factor.

A method has been developed to allow marking and removal of thin (<0.3mm) surface layers on complex thick-section parts, in order to allow removal of damaged material prior to re-peening, and has potential for wider application in the life extension of fatigue-damaged aluminium alloy structure.

### 8.4.6 Scatter Reduction in Fatigue Testing (G Clark and A F Cox - AMRL)

Evaluation of the effect of fatigue parameters such as spectrum type has conventionally relied on the use of multiple specimen tests to obtain a statistical estimate of fatigue life. One of the major sources of variability in specimen fatigue life, particularly at low load amplitudes, is the surface initial defect size and nature. An alternative approach being explored is the use of crack growth analysis on a much smaller number of components; this allows the determination of, say, the effects of spectrum type on fatigue crack growth, without the initial defect condition and the very early stages of crack propagation, both of which introduce high levels of variability, specimen-to-specimen. In this way, the number of tests required can be reduced.

### 8.4.7 NDE of Corrosion (G Clark and S R Lamb - AMRL)

The inevitability of the onset of corrosion, particularly in ageing aircraft, and the problems associated with its detection have been summarized in earlier reviews. This discussion highlighted the difficulty of NDI of multi-layer structures and the sometimes confusing effect of local geometric details, and evaluated the usefulness of various NDI methods. Specimens representing a large transport aircraft spar cap and web were prepared, containing flaws representing corrosion defects of various geometries. Examination of these specimens using Compton backscattered X-rays demonstrated an ability of the system to detect the larger flaws, but also indicated a number of access difficulties which limited the usefulness of the method. Preliminary trials of thermographic methods did not reveal any major indications of the flaws.

### 8.4.8 Acoustic Emission under Spectrum Loading (G Clark & P K Sharp)

Acoustic emission methods continue to be used on full-scale bulkheads being fatigue tested at AMRL. In the first test, an increase in AE output was noted at the time the crack was growing, from the region in which a crack was found. Laboratory tests conducted on the same material have been used to determine the AE output of the material during a fatigue test, and extensive noise reduction methods were employed to maximise detectability of the AE. The results indicated AE associated with five different regions of the fatigue load cycle. The principal regions of the load cycle were the peak load region, low and medium rising load portions, and to a lesser extent, the medium and low loads on the closing part of the cycle. The mechanisms associated with these outputs are consistent with crack face unsticking, and crack face asperity rubbing, and a peak load mechanism associated with crack advance.

The use of simple overloads produced high levels of AE output when the crack was advancing in a tearing mode (as has been observed in earlier work), although the precise mechanism has not yet been established positively. A major conclusion, however, was that for much of the crack growth, the AE output was principally associated with crack face asperity rubbing, and as such was extremely history-dependent. A single high load could change the output over subsequent cycles, as might be expected for this mechanism, and the AE approach is therefore inherently unreliable in the sense that its performance at any specific crack length is dependent on the precise details (even down to single loads) of the crack loading history of the component.

### 8.4.9 NDE of Small Fatigue Cracks (G Clark, P K Sharp & K Davey - AMRL)

Multiple cracking in flat surfaces, with final failure caused by the progressive linking of smaller cracks is a recognised feature of fatigue in the highly-optimised high-stress airframe components. AMRL has been trialling a variety of NDE methods to determine their ability to detect small cracks occurring in a flat surface. The trials indicated promising results for a system known as the Structural Integrity Monitor. This involves applying a polymer patch to the surface; the contact surface of the patch has fine channels which contain vacuum or atmospheric pressure, in an alternating arrangement. When a crack grows in the surface of the specimen, air leaks into the vacuum channels, and is detected by a leak rate sensor. The system could be used as a one-off inspection system, or be attached permanently as a continuous monitor, or as a permanent sensor which is interrogated only infrequently.

Trials showed the system could detect and located cracks of 0.5-1mm surface length under the sensor, and could also detect similar cracks in specimens which had been peened—a particularly difficult task for conventional NDI methods. Further trials are continuing.

### 8.4.10 Investigation of Fractal Dimensions in Fatigue Crack Growth (G Clark and N T Goldsmith - AMRL)

Fatigue crack growth in 7050 aluminium alloy specimens is being examined to determine whether fatigue progression features any evidence of preferred values. Such fractal dimensional behaviour has been suggested by Shanyavsky [A.Yu.Sasov and A.A. Shanyavsky, (1987) Fourier-fractography foundation of quantum mechanical nature of crack growth . Acta Stereol.6(3) 925-930], and would be expected to be a factor in the more detailed crack growth assessments involved in a range of current structural integrity problems. Future work is expected to examine a material which has a simpler microstructure in order to minimise variability in crack striation spacing.

### 8.4.11 Durability and Damage Tolerance of Bonded Composite Repairs (Richard Chester-AMRL)

As bonded composite repairs are considered for more demanding applications including repairs to primary structure, there is a need for greater understanding of the behaviour of the repaired structure under applied loads and the expected operating environment. The damage tolerance of the repair system is important as it is necessary to show that the crack growth rate has been reduced sufficiently to enable safe aircraft operation. The ability to predict accurately any crack growth is required to initiate an appropriate set of inspection intervals for the repair. A complicating fact is that the repair system itself is liable to damage as crack growth in the underlying structure occurs. Any damage to the repair system can reduce the repair efficiency thus complicating the prediction of subsequent crack growth rate.

Work has proceeded over the past year on crack patched panel test coupons to test the capability of a disbond model which is an extension of Rose's original patching model. This model takes into account the adhesive disbond which accompanies continued crack growth in the metallic structure, and has been shown to predict the correct form of the a versus N crack growth relationship. The effects of stress range and of varying patch thickness are also correctly predicted by the model. The behaviour of the test coupons to changes in R ratio suggest an insensitivity to R which is perhaps unexpected but which would greatly simplify the problem of predicting crack growth behaviour under spectrum

loading. The influence of temperature is quite complex as it influences a range of different properties. Currently the model appears to overcompensate for high temperatures although this may be due to inadequate adhesives allowables.
The disbond model has proved capable of predicting the behaviour of cracked patched test coupons representative of repaired aircraft structure under a variety of constant amplitude load situations. The long term goal is to extend the model to predict accurately the growth of cracks under spectrum loading.

### 8.4.12 Through-Thickness Stresses in Adhesive Joints ( Richard Bartholomeusz - AMRL)

Through-thickness stresses in bonded joints, that is stresses normal to the plane of the adhesive layer, generally arise from out-of-plane loading in the joint. These stresses are called peel, tension, tearing or normal stresses and usually result from the shear stresses developed during load transfer and from the development of secondary bending under non-axial loading. Less commonly they arise in joints with a high degree of curvature. Normal stresses play no part in the load transfer in the joint, but can cause failure since adhesives (and composite adherends) have relatively low peel strengths.

Currently two major repairs or reinforcements are being developed for the Royal Australian Air Force (RAAF) both of which are limited by the existence of through-thickness stresses resulting from curvature. These applications are the reinforcement to the F/A-18 470.5 bulkhead crotch region and repair/reinforcement of cracked F/A-18 aileron hinges.

In support of these repairs a program is underway to investigate the fatigue and static performance of adhesive joints when subjected to through-thickness or normal loading. A representative structural detail test specimen the "Curved Beam Specimen" has been designed to identify the stress state in a joint where the adhesive is loaded normal to the plane of the bondline. An experimental program to determine failure loads and durability has been used in conjunction with FE analysis, to determine the through thickness stresses in adhesives. Such specimens were tested both statically to failure and under spectrum fatigue loading to determine adhesive life. It is hoped that the CBS may prove to be an excellent specimen for providing generic data on through-thickness static and fatigue strength of adhesives.

### 8.4.13 Effect of plate thickness on the in-plane and through-thickness stresses at a hole (Rebecca Evans - AMRL)

Theoretical studies have indicated that the stress concentration factor (maximum local stress/remote stress) of a hole varies through the thickness of a plate and is sensitive to the value of hole diameter to plate thickness ratio (D/t), and to Poisson's ratio. Theory has also shown that the assumption of plane stress is adequate for thin plates which have a thickness less than one quarter the hole diameter. For thick plates—which have a thickness of at least twice the diameter of the hole—the three-dimensional stress concentration factor should be used.

To assess the validity of the various solutions, an experimental study, including thermoelastic and strain-gauge measurements, has been carried out at AMRL, [31]. The study examined the influence of plate thickness on the three-dimensional stress distribution of aluminium alloy specimens with central holes which were either reamed or cold expanded by 3.6%. The variations in surface stresses and through-the-thickness stresses, with specimens of thickness 10, 20 or 35 mm and hole diameter of 22.6 mm, have been investigated. Three-dimensional elastic finite-element analyses were also conducted.

The experimental data compared well with the numerical finite-element work for the surface of the specimens. The corresponding through-thickness data were not as clearly in accordance due to experimental problems. The numerical finite-element work which was undertaken validated the trends of other published finite-element and analytical studies, that as the thickness of a plate with a hole increased, the maximum stress concentration factor which is in the middle of the plate, increased and the minimum value of stress concentration factor which is at the surface of the plate, decreased, refer to figure 8. Thus, care must be taken in engineering calculations for thick plates with holes—the

common use of a two-dimensional handbook stress concentration factor may be too conservative at the surface, but unconservative at mid-thickness.

### 8.4.14 Effectiveness of high modulus interference-fit bushes for fatigue life extension of plates with circular holes (Rebbeca Evans - AMRL)

Theoretical and experimental investigations have respectively predicted and confirmed that the fatigue life of a plate with a hole can be increased by the use of an interference-fit insert. Some theoretical studies have considered inserts of various materials and wall thickness. However, very limited experimental verification has been given in the literature to assess the potential benefit of using various materials and wall thickness for the inserts.

A testing program to experimentally investigate the effect of the modulus of an interference-fit bush on the fatigue life of an aluminium plate with a central hole (of 12 mm diameter) has been carried out at AMRL, [32]. The study examined specimens with 1 mm or 2 mm thick bushes of 0.5% interference fit and of different materials; namely aluminium, steel and tungsten carbide. The specimens were fatigue tested under a representative fighter-aircraft loading sequence. The thin-walled bushes extended the fatigue lives of the plates with holes by at least 4.2 times, and the thick-walled bushes produced at least a 10.9 times increase in life, refer to Table 1. The test results also indicated that different failure modes can occur depending on the relative thickness and modulus of the interference-fit bush. The specimens with the highest bush modulus did not necessarily have the longest lives, even though the stress concentration factor due to remote loading at the hole edge for these specimens was the lowest.

Finite-element work was also conducted to determine the stresses and strains in the interference-fit bushes for conditions of no-slip and frictionless contact between the bush and uniaxially loaded plate. High tensile strains were found to occur in the bush at the inner surface.

| Bush | Thin-walled bushed specimens log. average fatigue life (flight hours) | Thick-walled bushed specimens log. average fatigue life (flight hours) |
|---|---|---|
| Tungsten-carbide | 45367 | 162834 |
| Steel | 37393 | 215270 |
| Aluminium | 24949 | 431313 |
| Hole only | 5865 | 14948 |

**Table 1. Spectrum fatigue lives of bush modulus specimens**

### 8.4.15 Reduction In Adhesive Shear Strains At The Ends Of Bonded Reinforcements (T. Tran-Cong and M. Heller - AMRL)

There are a number of outstanding issues concerning the improved design of bonded repairs which need to be addressed. One such issue is the large peak in the adhesive shear strain which typically occurs near the *end of the patch*, which can cause failure of the adhesive system and compromise the performance of the repair. To reduce the severity of this peak, uniform stepping of the multilayer patch is currently being employed for bonded repairs undertaken on RAAF aircraft, with the typical step length being 3 mm per lamina ply. However, this uniform step length is not optimal and it is believed that a much better option is to design a non-uniform laminae stepping arrangement. The configuration under study is shown in Figure 9. A cracked plate of uniform thickness $t_i$ is loaded by a remote stress $\sigma_O$. A stepped composite patch is bonded to each side of this plate by an adhesive layer having a uniform thickness $\eta$. The maximum thickness of each patch is denoted $t_O$. Each patch is stepped at each end identically, and each of the n steps is in general allowed to be of different thickness, modulus and arbitrary (i.e. non-uniform) length.

In this work, a one dimensional theory is developed to minimise the maximum shear strain in the adhesive, thereby leading to an improved design for multi-layer patches. The approach used is to allow each step to be of different thickness and modulus, and of variable step length, while keeping the total length of the patch unchanged. It is shown that, to achieve a 20% reduction in peak adhesive shear strain for a typical unidirectional stepped patch, the first step adjacent to the patch edge needs to be much longer than the remaining steps. For the case where cross-ply laminae are used in conjunction with unidirectional laminae, the maximum shear strain in the adhesive layer can be reduced by about 60%. For example, consider the adhesive shear strain distribution in the stepped region for the case of a typical bonded repair (consisting of 10 equal step heights of 0.13 mm, and a uniform step length of 3 mm) and given in Figure 10. We can contrast this with the adhesive shear strain (g) distribution for a non-uniform stepping case, with 11 unequal step thicknesses with non-uniform step lengths, (consisting of a combination of 10 cross-ply and 9 unidirectional laminae), Figure 11. It can be clearly seen that the peak strain value has been reduced by about 60 % from that given in Figure 10. The results obtained in this work indicate, as expected, that reduced peak adhesive shear strains lead to a smoother transition of load from the plate to the patch. This suggests that a shear-stress optimised patch design will also reduce the undesirable stress concentration in the repaired structure

### 8.4.16 Reduction Of Plate Stress Concentration Factors Due To Bonded Reinforcements (L. Shah, M. Heller, C. H. Wang, AMRL And J. F. Williams Melbourne University)

Components repaired or reinforced by bonded composite patches experience elevated stresses near the end of the patch. This stress elevation may be minimised by the correct design of the local geometry and material properties near the end of the patch. For example, this can be achieved by appropriate stepping of the patch, or through the addition of extra adhesive, in the form of a fillet, near the end of the patch.. The configuration under study is shown in Figure 9 and is as discussed earlier **in section 8.4.15.** It should be noted in related work, (section 8.4.15) a previous investigation was undertaken to determine an improved geometry at the ends of a stepped patch that would result in the maximum adhesive shear strains being lowered, [33]. Furthermore, in reference [29], an analytical method has been formulated to describe the stress concentration in the plate and to determine what type of shear stress distribution in the adhesive gives the best reduction in the plate stress concentration.

Hence in the present investigation [35], finite element analyses are used to determine the influence on the plate stress concentration for a bonded repair specimen of the following: (i) modification of the shear stress distribution in the adhesive by changing the distribution of the patch stiffness and stepping arrangement, and (ii) introduction of an adhesive fillet at the end of the patch, (see Figure 12). Analyses were undertaken for the following three patching cases: (i) patch with one step, i.e. a uniform thickness patch, (ii) patch with multiple uniform step lengths, and (iii) patch with multiple non-uniform step lengths, unequal step heights and different lamina moduli. For all cases the plate was aluminium alloy and the patch consisted of unidirectional or cross-ply boron/epoxy laminae. The finite element results obtained are summarised in Table 2. It can be seen that a reduction of the maximum stress

concentration factor from 2.22 for the single step (uniform thickness) patch case to 1.28 for the multiple step (with non-uniform stepping) patch case was achieved. This was further reduced to 1.17 with the addition of a suitable adhesive fillet. A recently postulated close relationship between the magnitude of the plate stress concentration and the peak adhesive shear stress [32,33] has been confirmed.

### 8.4.17 A Gradientless Finite Element Method For Determining Optimal Shapes To Reduce Stress Concentration Factors (R. Kaye and M. Heller - AMRL)

In the present work an efficient gradientless computational method has been developed that can be used with any standard commercial finite element package, for determining geometries with minimised stress concentrations, [36]. The driving force for the method is to achieve constant boundary stress by correctly moving boundary nodal positions. The approach undertaken can be summarised as follows. Consider a plate containing a stress concentrator such as an arbitrary shaped hole. In the finite element discritisation of this geometry, there will be a set of *i* nodes on and around the hole boundary. When the plate is under the action of arbitrary remote loading, the tangential (ie hoop) stress at these nodes will vary in magnitude around the boundary. As shown by previous work in the literature, the magnitude of the stress concentration at theses positions, (ie nodes) can be reduced by adding material where stresses are high and/or removing material where stresses are low. The question is then what amount of material should be added or removed. A simple postulate would be to require the amount of material added or removed at any point on the boundary to be directly proportional to the difference between the local tangential stress and a reference value, and this process being repeated iteratively until a reduced stress concentration has been achieved. Hence the amount to move a given node *i* on the boundary, (in the normal direction) can be written as,

$$d_i = \left( \left( \sigma_i - \sigma_{th} \right) / \sigma_{th} \right) \times r \times s$$

where positive $d_i$ indicates material removal and negative $d_i$ indicates material addition,

and

$\sigma_i$ = *tangential stress at node i on the boundary*

$\sigma_{th}$ = *threshold boundary hoop stress*

$r$ = *arbitrary scaling distance, such as initial hole radius.*

$s$ = *arbitrary step factor typically in the range 0.1 - 0.4*

Appropriate selection of the parameter $\sigma_{th}$ determines whether material can be either added or removed around the boundary. This iterative algorithm has been automated in conjunction with the finite element code PAFEC, and the method has been demonstrated on a number of benchmark problems. For illustrative purposes, consider the case of a biaxially loaded plate containing a square hole as shown in Figure 13, with the aim of changing the shape of this hole to minimise the magnitude of maximum stress on the boundary. The correct solution of an elliptically shaped hole, with a constant tangential stress around the boundary of 165 MPa, was achieved after about 18 iterations, as shown in Figure 14. With some slight modification, the method has been also applied to the problem of determining a improved tapering at the end of aluminium doublers bonded to an aluminium plate, subjected to uniaxial loading. The aim was to minimise the peak adhesive stress by changing the profile of the tapered region. The initial and final finite element geometries in the tapered region are shown in Figure 15, and the corresponding adhesive shear stresses are given in Figure 16.

| Cases considered | Adhesive shear stress (MPa) | Plate stress concentration |
|---|---|---|
| **Patch with single step** | | |
| *No fillet* | 26.9 | 2.22 |
| *Rectangular fillet cases* | | |
| 0.4 mm length | 20.9 | 1.76 |
| 1 mm length | 20.3 | 1.45 |
| 2 mm length | 20.0 | 1.42 |
| 12 mm length | 19.6 | 1.42 |
| **Patch with uniform stepping** | | |
| *No fillet* | 14.7 | 1.65 |
| *Rectangular 2 mm fillet length* | | |
| Adhesive level fillet height | 11.8 | 1.34 |
| First step fillet height | 11.2 | 1.26 |
| Second step fillet height | 11.1 | 1.24 |
| Every step fillet height | 11.0 | 1.21 |
| **Patch with non-uniform stepping (with cross-ply lamina)** | | |
| *No fillet* | 5.6 | 1.28 |
| *Rectangular 2 mm fillet length* | | |
| Every step fillet height | 4.9 | 1.17 |

**Table 2. Adhesive shear stress and plate stress concentrations for various stepped patch cases.**

### 8.4.18 Durability of Postbuckling Stiffened Fibre Composite Shear Panels (Rodney S. Thomson[1], and Murray L. Scott[2])

An investigation employing theoretical and experimental techniques is being undertaken to assess the performance of thin stiffened composite panels in the postbuckling state [Reference 37]. Shear testing was performed on three-blade-stiffened panels with overall dimensions of 340 x 340 mm and working dimensions of 250 x 250 mm. The integral blade-stiffeners were 25 mm high with run-out angles of 45°. The skins and stiffeners were co-cured in an autoclave using flexible tooling. The material used to manufacture the panels was T300/914C carbon fibre/epoxy, unidirectional, pre-impregnated tape. The skins were 1 mm in thickness, consisting of eight plies of tape, and the stiffeners were 2 mm in thickness, consisting of 16 plies. Panels were also manufactured with a 50 mm diameter teflon insert positioned under the middle stiffener to simulate the presence of a delamination, while other panels have been impacted to produce barely visible impact damage.

Static testing of undamaged panels showed that the panels buckled at approximately 88 N/mm developing a diagonal tension field typical of shear buckling. The presence of the 50 mm diameter teflon insert did not greatly influence the buckling load of the panels. However, the postbuckling shape was affected as one of the buckle peaks was elongated and clearly extended into the delaminated region. The undamaged panels failed at loads of approximately 226 N/mm which was preceded by audible cracking from a load of 175 N/mm. In the teflon insert panels, cracking was heard at a load of 107 N/mm while significant delamination growth was noted at a load of 136 N/mm. The presence of the delamination reduced the strength of the panels by 25%. The prediction of delamination growth is currently being modelled using the finite element method to determine strain energy release rates.

Laboratory fatigue tests have been conducted on several panels. One test was carried out for three million cycles with a minimum load of 57 N/mm and a peak load of 141 N/mm. No resulting damage was detected by ultrasonic C-scan. Another test was carried out from -141 to 141 N/mm. This panel was subjected to 60,000 cycles with no resulting damage.

Long term environmental fatigue and static exposure testing was performed at the Australian Defence Science and Technology Organisation tropical test facility located near Innisfail in Northern Queensland. The specimens underwent tropical exposure for up to 20.5 months during which time the various fatigue specimens were subjected to a total of 370,000 cycles. All specimens exhibited signs of degradation due to exposure to UV radiation, characterised by the breaking down of the outer layer of the epoxy matrix which was then eroded by the wind and rain. The undamaged fatigue specimen exhibited no signs of damage development while the impact damaged panel showed insignificant growth. The specimen with the artificial delamination did undergo delamination growth. This was clearly visible after 91,000 cycles by C-scan. Another delamination specimen was tested for 112,000 cycles and suffered similar delamination growth. C-scans before and after testing of a panel containing a teflon insert are shown in Figure 17. From the various C-scans of the panels containing artificial delaminations, it appeared that the delamination reached a limit after approximately 100,000 cycles. This testing program has demonstrated the inherent damage tolerance of the stiffened, postbuckling, composite panels.

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## 8.5 REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN NEW ZEALAND

### 8.5.1 A-4K Skyhawk (W. L. Price - DSE)

Requirements have been defined for a loads/environment spectrum survey (L/ESS) on RNZAF Skyhawk aircraft. The project is aimed at clarifying the remaining structural life of the A-4K which is a modified version of Skyhawk aircraft in service elsewhere in the world. Project definition was based on a review of design and fatigue test data and on information from RNZAF maintenance authorities (Reference *38*). Initial evaluation concluded that data from the full scale fatigue tests have only limited relevance to the current configuration of New Zealand aircraft. Past cases of wing fatigue cracking in the RNZAF Skyhawk fleet indicated that damage rates in these aircraft were high. A load/environment spectrum survey with emphasis on strain measurement is needed to evaluate fatigue damage rates in critical structural components. Final preparations for the project are now under way.

### 8.5.2 Aermacchi MB-339CB (P. J. Riddell - DSE)

A review has been made of the technical requirements for structural life management in RNZAF MB-339 aircraft. Airframe monitoring has indicated that fatigue life consumption rates are higher than predicted. Existing aircraft tracking methods have not been able to determine the reasons for the accelerated damage rate and a small-scale loads/environment spectrum survey has therefore been instituted to evaluate individual missions and manoeuvres. Initial indications are that design assumptions correctly support the existing method of estimating of fatigue damage in service. There are no indications that manoeuvre load factors are more severe than predicted so the frequency and stress range of manoeuvres are being investigated. Time history data from operational flying is being dissected to clarify these effects (Figure *18*).

### 8.5.3 Structural Data Recorder (P. J. Riddell - DSE)

The MB-339 L/ESS described above is based on the Networked Airframe-Data Acquisition and Recorder System (NADARS). The system, which is based on a distributed network of compact microprocessors, has been developed to provide measurements of fatigue damage in RNZAF aircraft. NADARS recorders, which can can be installed quickly and with simplified wiring requirements, provide structural and parametric data in time sequence format. Instantaneous fatigue damage rates can therefore be correlated with individual missions and manoeuvres.

### 8.5.4 Viper 680-43 Zero Stage Compressor Blade HCF/FOD Damage Tolerance (P. C. Conor - DSE)

The loss in 1993 of an MB-339 because of a compressor blade failure prompted a detailed investigation of high cycle fatigue (HCF) and foreign object damage (FOD) tolerance. Conducted as a joint project between the RNZAF and Rolls-Royce, the project supported the development of a damage tolerant blade to meet New Zealand operating requirements. The New Zealand segment of the work included an assessment of the geometry of foreign object damage induced in service. Artificial FOD defects were also evaluated in detail. The development programme included an optometric study (by Auckland University) of the visibility of small edge defects, an assessment of FOD detectability on the flight line and measurements of engine use in routine operational service (Reference *39*). The work also involved development of a severely-notched (Kt7) high cycle fatigue specimen notched by abrasive cutting instead of electrical discharge machining. The results of the project assisted with the development of an altered compressor blade which has been accepted into service by the RNZAF. Residual questions about the effect of small FOD on fatigue crack initiation and growth are being addressed in further research work.

### 8.5.5 Aging Engines: J-52 Engine Oil Tube Cracking
**(P. C. Conor, N. F. Ellis, A. D. James - DSE)**

The recent loss of an RNZAF A-4K Skyhawk aircraft was found to have been caused by corrosion pitting and fatigue cracking in lubricating oil pipelines around the engine. Similar cracks was were found to be widespread in other engines in the fleet. Corrosion, stress corrosion and corrosion fatigue damage were found to have been accelerated by antimony chloride formed by reaction between fire retardant chemicals in tape used to reduce fretting damage on the pipework. Laboratory tests confirmed the extreme sensitivity of the stainless steel to attack by antimony chloride. Antimony compounds released by hydrolysis of fire retardant polymers appear to have the potential to accelerate corrosion and corrosion fatigue in a variety of aircraft systems.

### 8.5.6 Performance of Composite Laminates
**(D. P. W. Horrigan - University of Auckland)**

Impact tests have been carried out on thin skinned honyecomb sandwich laminates. The aim of the project is to enable commercial operators to assess the type and extent of damage of various types. The work is expected to investigate the spread of core and skin damage under various conditions of cyclic loading. Parallel work is aimed at developing non-destructive methods of assessing laminate quality. Vibration tests are being used to determine the elastic moduli of composite panels. Firstly, this enables the use of accurate elastic modulus values for finite element analysis. Secondly, the work performed so far indicates that there is a relationship between modulus and laminate strength. A related project has been concerned with the development of formed thermoplastic composite laminates to replace more expensive materials with inferior impact resistance.

### 8.5.7 Boeing 747-200 - Evaluation of Fuselage Strain
**(D. P. W. Horrigan - University of Auckland, M. P. Pervan - Air New Zealand Limited).**

In-flight strain measurements have been made to determine stress levels in skin and structural components at Section 41 in a Boeing 747-200. Finite element analysis of this zone of the aircraft is being performed to permit the measured stress to be compared with design data.

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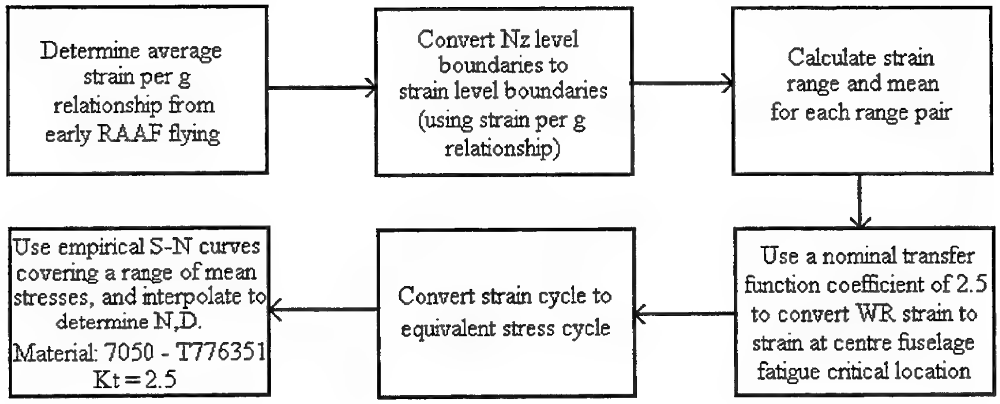



**Figure 1. Development of the Unit Damage Matrix**


**Figure 2. Boron Epoxy Patch Repair to RAAF F-111**

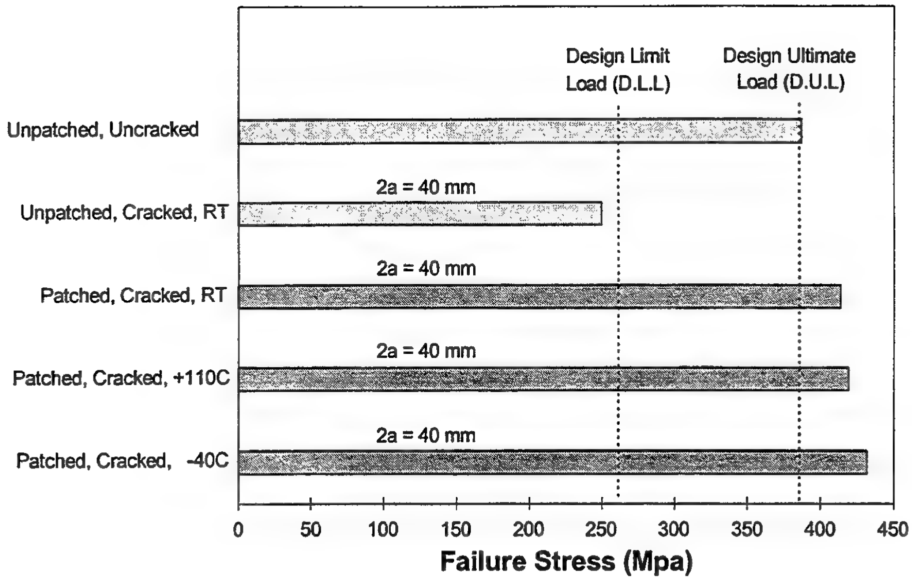



**Figure 3. Residual Strength Test Results, F-111 Bonded Repair**

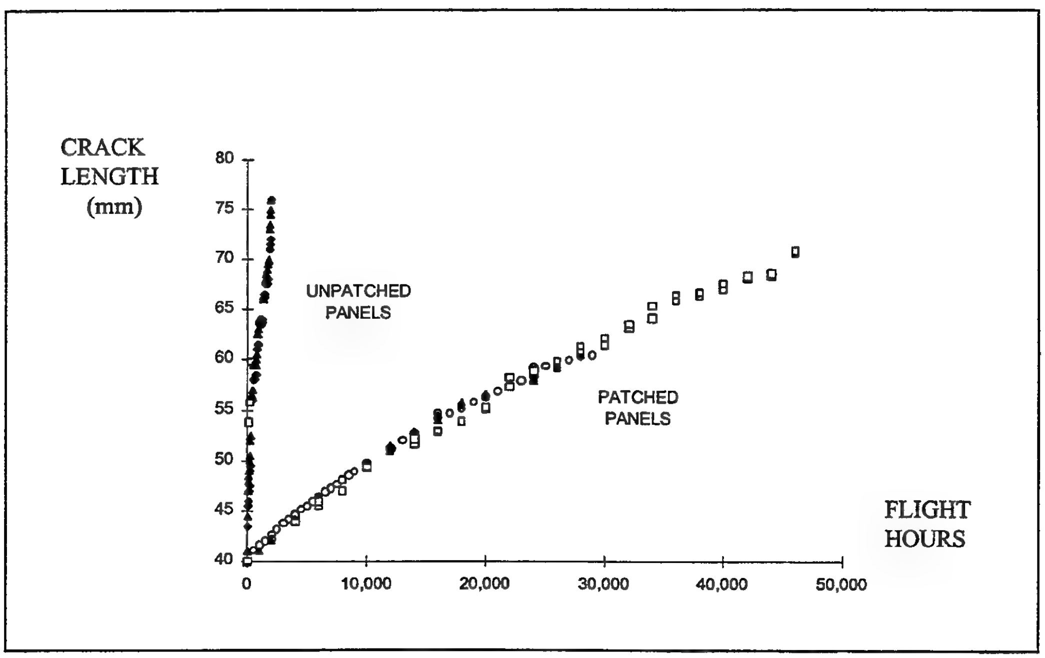



**Figure 4. Spectrum Load Crack Growth Results, F-111 Bonded Repair**

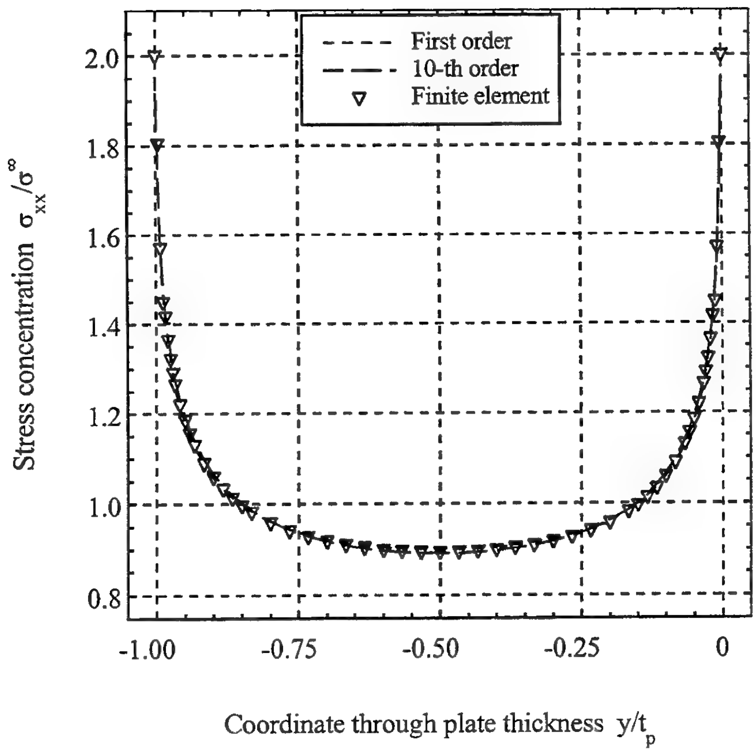



**Figure 5. Through thickness stress concentration in the inner adherend of double lap joint**

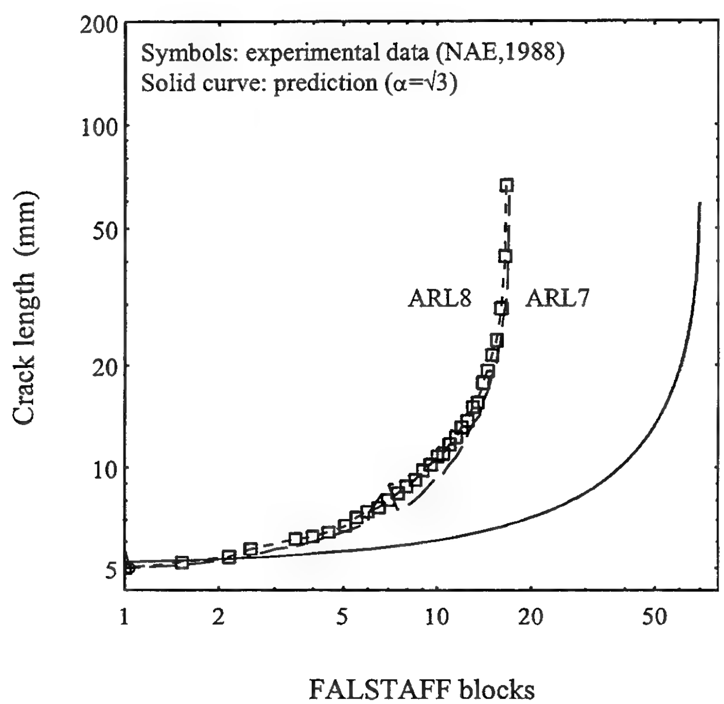



**Figure 6. Unpatched Al sheets with honeycomb core**

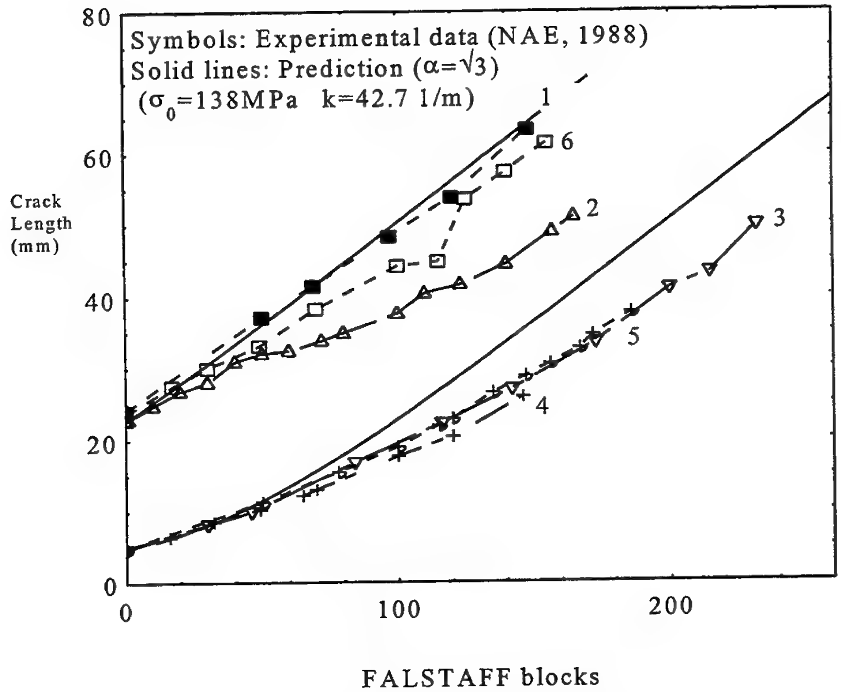



**Figure 7. Crack growth in patched panels**

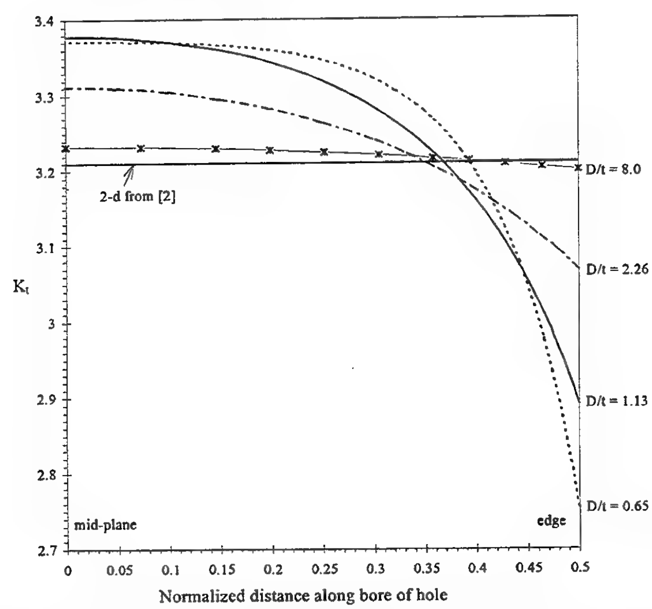



**Figure 8. Stress concentration factor along bore of hole for various hole diameter to thickness ratios.**

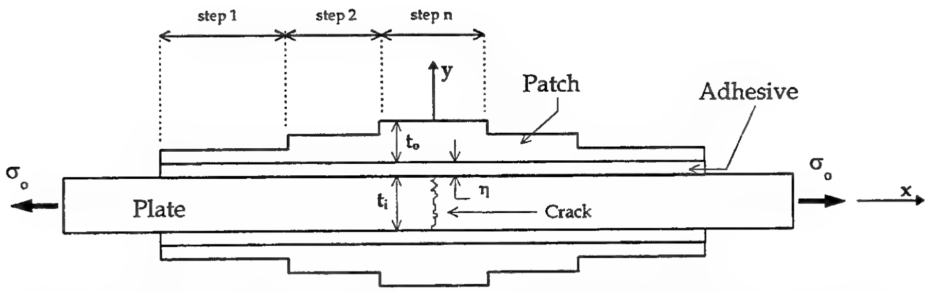



**Figure 9. Geometry of a typical plate reinforced with double-sided bonded stepped patches**

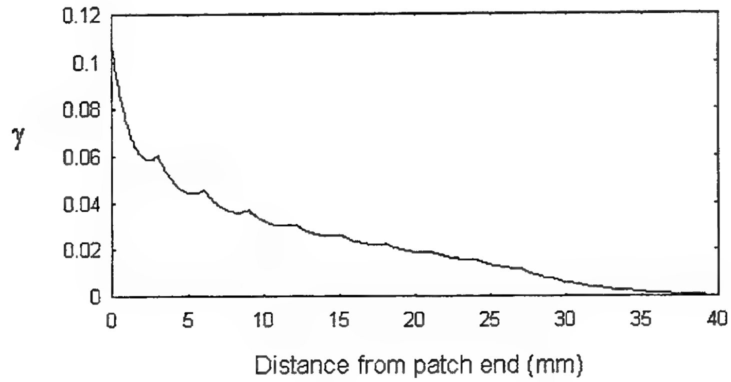



**Figure 10. Adhesive shear strain (g) distribution where the patch consists of ten steps of qual height with uniform step lengths of 3 mm.**

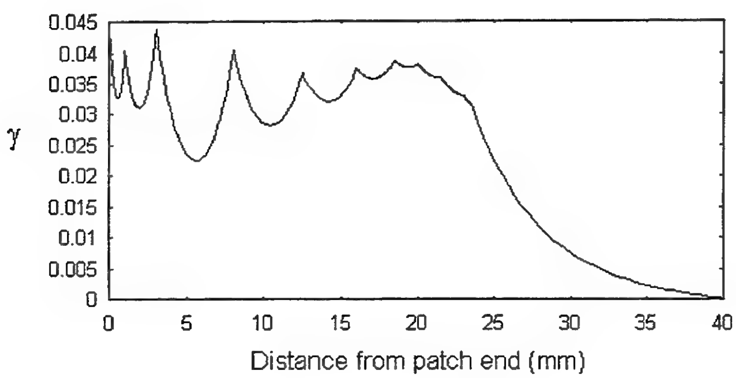
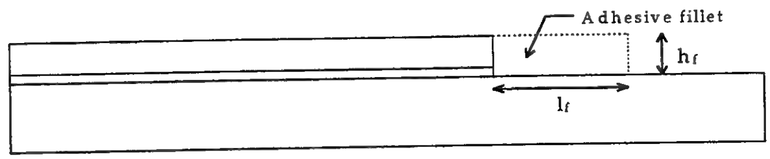



**Figure 11. Adhesive shear strain (g) distribution for the case of 11 unequal step thicknesses with non-uniform step lengths, (consisting of a combination of 10 cross-ply and 9 unidirectional lamina).**

(a)




(b)

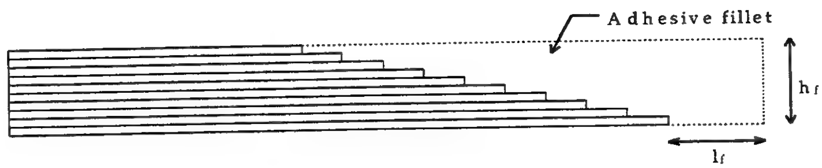



**Figure 12. Typical patch geometries with typical rectangular adhesive fillets shown; (a) Plate with single step patch, (b) Typical multiple uniform stepping arrangement of patch.**

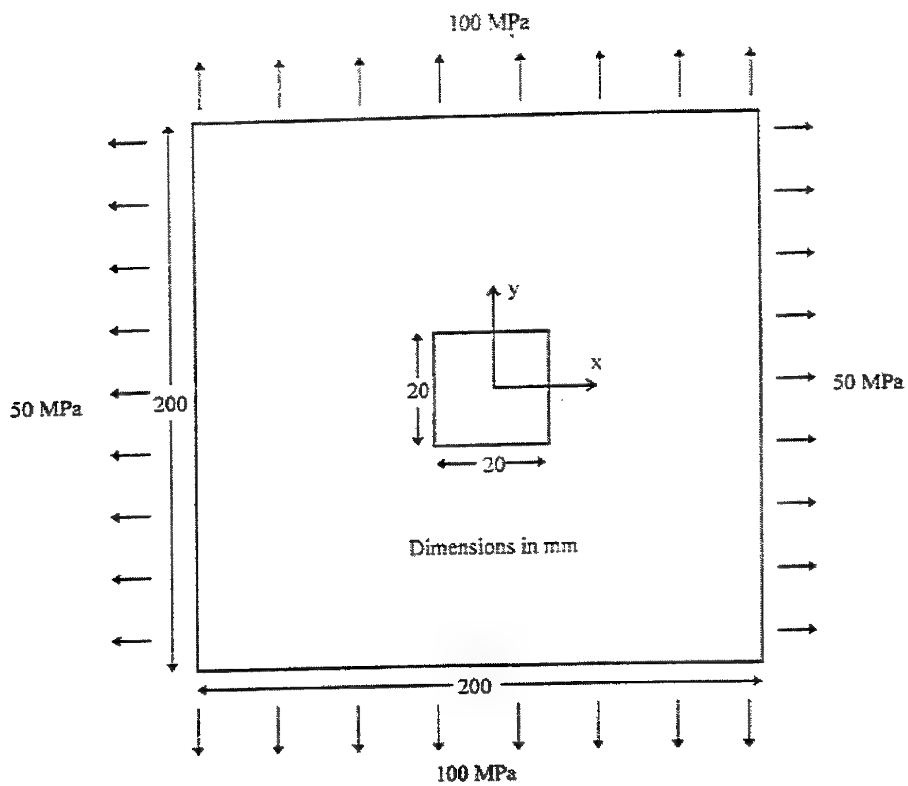



**Figure 13. Initial geometry and loading arrangement for a large square plate containing a square hole for a 2:1 biaxial stress field.**

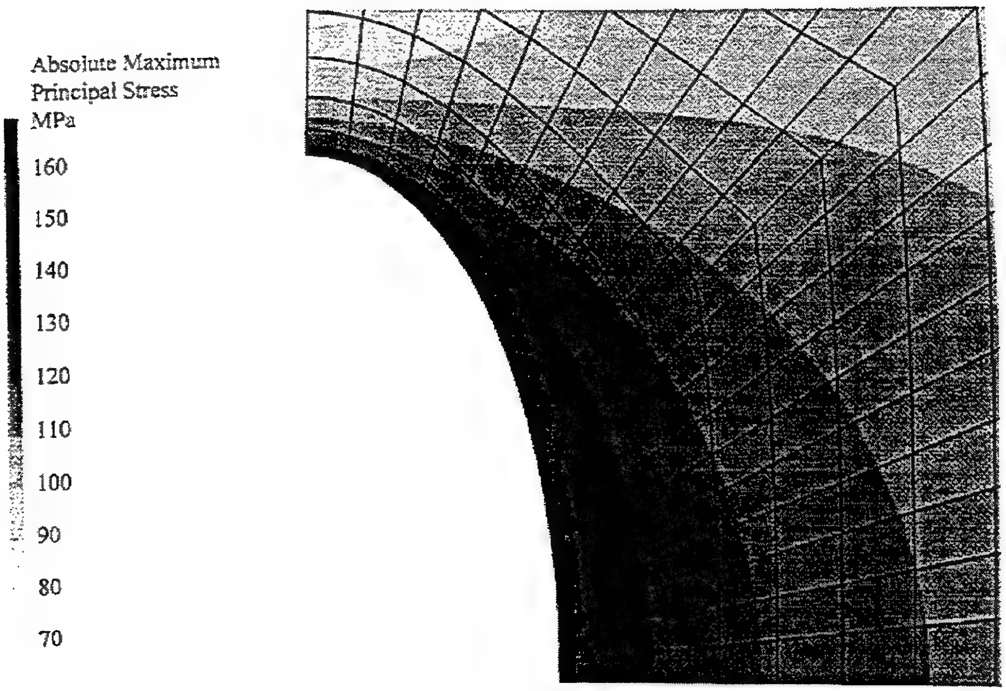



**Figure 14. Solution geometry and maximum principle stress distribution for final optimisation iteration for a large square plate containing an initial square hole for a 2:1 biaxial stress field.**

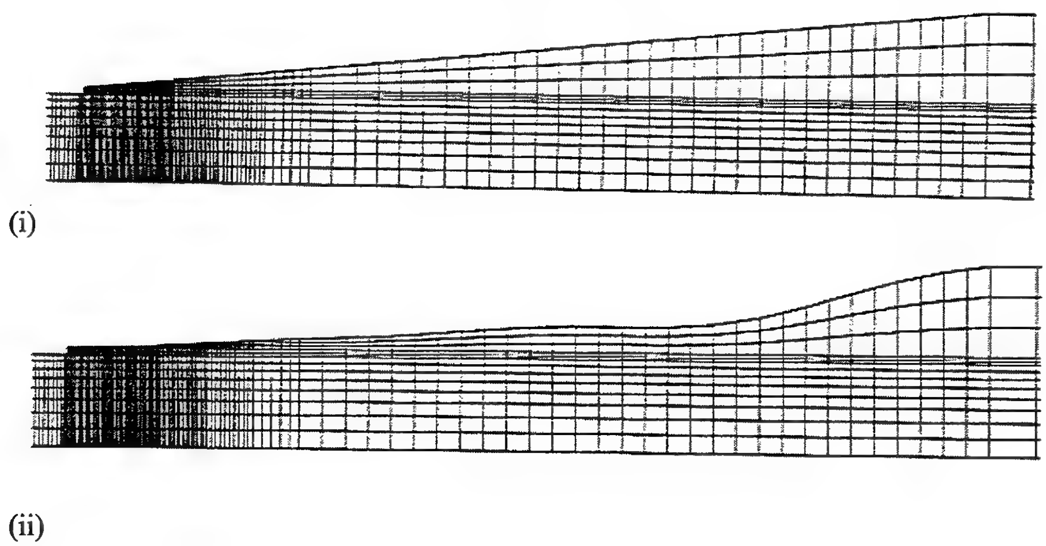



**Figure 15. Geometry of taper region for (i) initial and (ii) final optimisation iteration for a bonded double lap joint with a uniaxial applied stress**

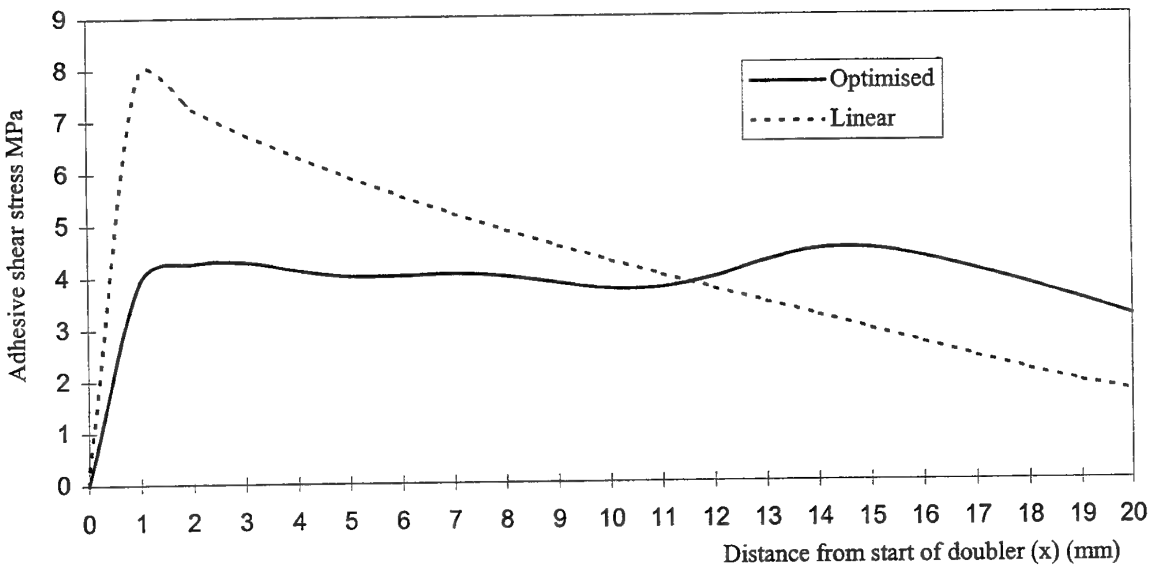
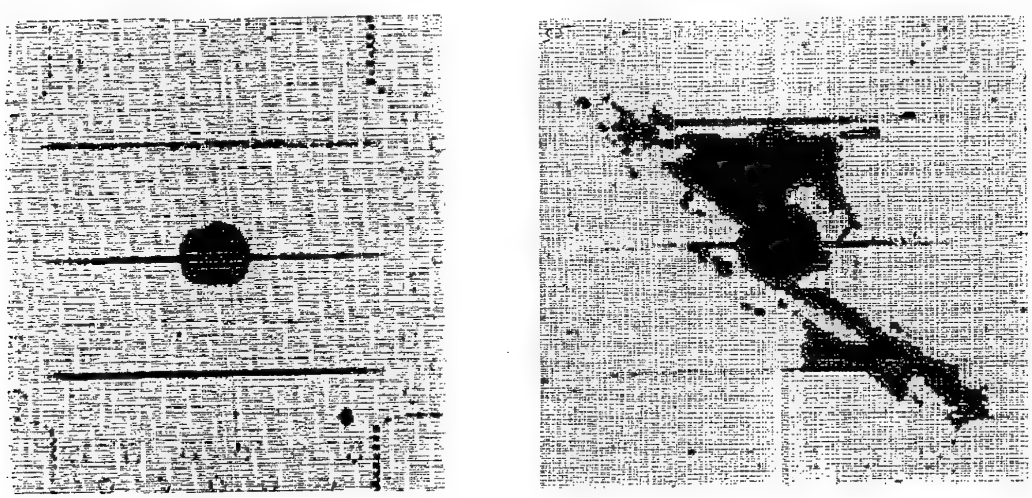



**Figure 16. Adhesive shear stress distribution for solution geometry of final optimisation iteration for a bonded double lap joint with a uniaxial applied stress (tapered region only shown)**



*(a)* *(b)*

**Figure 17. C-scans of teflon insert panel (a) prior to testing and (b) after 112,000 cycles.**

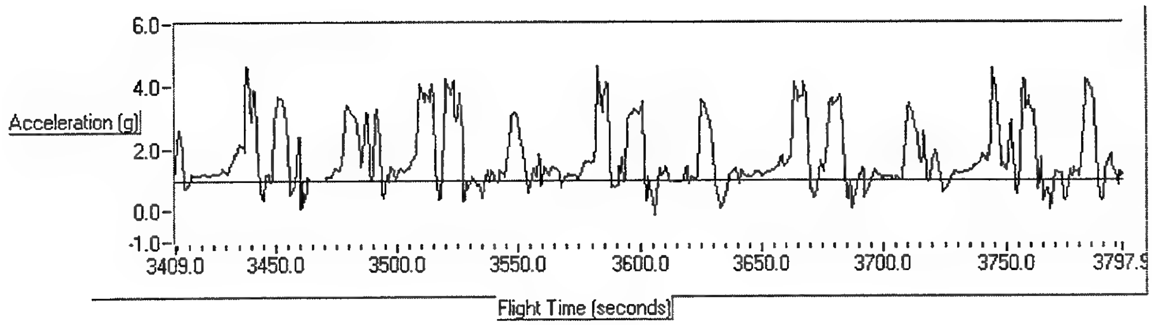



**Figure 18. A section of an operational loads recording from a training mission flown on an RNZAF MB-339 aircraft. The measurements, which were obtained using a NADARS recorder, provide information about the frequency and severity of manoeuvres in the current RNZAF pilot training programme.**

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19. ABSTRACT

This document is for presentation to the 25th Conference of the International Committee on Aeronautical Fatigue scheduled to be held in Edinburgh, Scotland on 16th and 17th June 1997. A review is given of the aircraft fatigue research and associated activities which form part of the programs of the Aeronautical and Maritime Research Laboratory, the Civil Aviation Safety Authority, and universities in Australia and the Defence Scientific Establishment and the University of Auckland in New Zealand. The review summarises fatigue-related research programs as well as fatigue investigations on specific military and civil aircraft.

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